TECHNICAL MEMORANDUMS

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 852

SYSTEMATIC AIRFOIL TESTS

IN THE LARGE WIND TUNNEL OF THE DVL

By H. Doetsch and M. Kramer

Luftfahrtforschung
Vol. XIV, No. 10, October 12, 1937
Verlag von R. Oldenbourg, München und Berlin

Washington
March 1938

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SYSTEMATIC AIRFOIL TESTS IN THE LARGE WIND TUNNEL OF THE DVL*

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SUMMARY

The present report is a description of systematic tests at maximum lift on airfoils with and without split flap and of profile drag at low lift. The program included, respectively, the symmetrical and 2-percent camber N.A.C.A. airfoil sections 00, 24, and 230, with 9- to 21-percent thickness range. The maximum lift of the airfoil series without split flap was established for the entire practical flying range by comparing the DVL data with the findings from other wind tunnels. In order to obtain an opinion as to the suitability of the airfoils with flaps, the maximum-lift measurements were repeated on airfoils with split flaps.

The profile drag at low lift was arrived at by direct weighing and momentum measurements and, since the profiles were of unusual depth, extended to large Reynolds Numbers. It results in very carefully developed curves $c_{a_{max}}/c_{wp}(c_a = 0.1)$ with and without split flap, which as regards Reynolds Number correspond to actual flight conditions.

I. INTRODUCTION

As the 5- by 7-meter wind tunnel of the DVL did not begin to operate until in the fall of 1935 (reference 1) only the utmost restrictions in the scope of the research program made it possible to catch up with other countries which were years ahead. For this reason only two airfoil

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*"Systematische Profiluntersuchungen im grossen Windkanal der DVL." Luftfahrtforschung, vol. 14, no. 10, October 12, 1937, pp. 480-485.

series significant for the future were chosen from among the many potentialities and investigated very painstakingly.

The choice fell to the symmetrical airfoil series with from 9- to 21-percent thickness, which correspond to N.A.C.A. airfoil sections 0009 to 0021, and to the series with 2-percent camber at 40-percent chord and 9- to 21-percent thickness, which correspond to the N.A.C.A. series 2409 to 2421 (reference 2). In the meantime, U. S. investigations had shown that a forward shift of the 2-percent camber to 15-percent chord insured a further improvement in the airfoil sections (reference 3). And this fact prompted the inclusion of three airfoil sections of the N.A.C.A. series 23009 to 23021.

Almost even more essential than the investigation of ordinary airfoils seemed the elucidation of the question as to what airfoil was best suited in conjunction with a landing aid at the trailing edge. In this connection, it was necessary to select a landing aid which combined great effect with little Reynolds Number sensitivity as well as easy installation on any airfoil section. A split flap extending over the entire span was chosen, because it has clear separation edges and is therefore less responsive to Reynolds Numbers.

The result of this test series is precisely valid for the split flap. But, since the reaction of the various otherwise customary landing aids on the leading edge is intimately related, the result applies to ordinary flaps and split flaps as well, at least as a first approximation. Ostensibly the subsequent supplementary inclusion of perceptibly different landing aids, such as Fowler flaps, in the test program is necessary.

II. EFFECTIVE REYNOLDS NUMBER AND TURBULENCE FACTOR

The 5- by 7-meter tunnel of the DVL was designed with a view to minimum jet turbulence. This aim proceeded from the knowledge that atmospheric turbulence is proven to be very small (reference 4) and a clear concept of the manner in which the turbulence changed the airfoil characteristics did not exist. The turbulence of the DVL tunnel is, in fact, very low. A sphere with a maximum critical Reynolds Number of 4.05×10^5 in flight in still air manifested the value 3.7×10^5 in tunnel center.

In the meantime the Americans fortunately succeeded in proving by comparison of sphere and maximum-lift measurements in the N.A.C.A. variable-density tunnel and in the N.A.C.A. full-scale tunnel, that the turbulence lowers the critical Reynolds Number of the sphere in the same ratio as it does for the maximum-lift measurements, (reference 5), or in other words, that for maximum-lift investigations the Reynolds Number of the test must be multiplied by the ratio of the critical Reynolds Number of the sphere in nonturbulent air to that in the tunnel in order to obtain the Reynolds Number of the maximum lift measurement applicable in flight. The ratio of the critical Reynolds Number of the sphere in nonturbulent air stream to the critical Reynolds Number in the tunnel is called the "turbulence factor."

$$\text{T.F.} = \frac{\text{Reynolds Number}_{(\text{still air})}}{\text{Reynolds Number}_{(\text{tunnel})}}$$

(Reynolds Number of sphere for drag $(c_W) = 0.3.$) The Reynolds Number which is valid for maximum lift in flight and which is obtained by multiplying the turbulence factor with the Reynolds Number of the maximum-lift measurement is called "effective" Reynolds Number.

$$R_{\text{effective}} = R_{\text{test}}\ \text{T.F.}$$

The effective Reynolds Number has proved satisfactory in the comparison of $c_{a_{max}}$ measurements effected in several different tunnels as well as in free flight (reference 5). It constitutes a definite advance in the elucidation of the $c_{a_{max}}$ question and removes the existing uncertainty. It is used hereinafter for comparing the $c_{a_{max}}$ measurements of the DVL with those of other tunnels.

The turbulence factors of various tunnels are listed in table I.

Table I

Tunnel	Turbulence factor	Source
5 × 7 m DVL	1.1	reference 1
1.5 m NACA (VDT)	2.64	" 3
3 m GALCIT	1.1	" 5

III. RESULTS OF $c_{a_{max}}$ MEASUREMENTS WITHOUT SPLIT FLAP

The airfoil models, consisting of a steel framework covered on a drawbench with a 10-millimeter layer of marble cement, had 4-meter span and 0.8-meter chord; the surface was smooth and highly polished (fig. 1).

The wing tips were rounded off, since the customary blunt tips result in appreciable errors which change with the airfoil thickness (reference 6). The rounding was so effected that the radius of rounding corresponded at each point of the profile chord to half the local profile thickness. The effect of this rounding on the maximum lift was investigated (fig. 2). It was found that for the practical thickness range the maximum lift drops about 3 percent, unaffected by the thickness. Since the rounded tips removed an essential error in the profile drag measurements, while its effect on the maximum lift is minor and not affected by the thickness, it was employed throughout the test program and corrected with the factor 0.97 in the comparison of the maximum lift of other tunnels.

Figures 3 to 5 illustrate the $c_{a_{max}}$ test data of the DVL tunnel for the series 00, 24, and 230, plotted against the effective Reynolds Number. For additional explanation the wing chord for approximately 100 kilometers per hour landing speed has been included.

The comparative data from the California Institute of Technology tunnel (GALCIT) and from the N.A.C.A. variable density tunnel (VDT) were selected for the following reasons:

The GALCIT is a 3-meter tunnel with closed experiment chamber which for low turbulence corresponds to the DVL tunnel and, as regards the obtainable Reynolds Numbers, extends the DVL measurements in the direction of lower Reynolds Numbers. The comparison with this tunnel was to prove the reliability of the DVL tunnel at low speeds. Unfortunately, only one test series of this tunnel is known (reference 7).

The N.A.C.A. VDT is a 1.5-meter high-pressure tunnel with closed test section whose jet is very turbulent (T.F. = 2.64). This tunnel was included in our comparison because with its effective Reynolds Numbers it extends the DVL measurements toward larger R and at the time is the tunnel in which the most extensive systematic measurements have been made so far. With its maximum attainable Reynolds Number the N.A.C.A. VDT offers any amount of desirable data. On the other hand, only very little data on systematic tests with low Reynolds Numbers are available (references 3 and 8). The N.A.C.A. VDT data at low Reynolds Number have not been included in the figures 3 and 8 because they are not systematic and apparently disclose scattering. Adding this scarce material would prove nothing while detracting from the otherwise lucid representation.

Analysis of the entire data in figures 3 to 5 manifest the following:

1) The DVL findings agree with the unfortunately scarce result of the GALCIT (on airfoil section 2412). For effective Reynolds Number $\cong 1.5 \times 10^6$ the result in the DVL tunnel seems to be more reliable than that in the GALCIT. The somewhat too small results of the latter at its maximum Reynolds Numbers are probably due to the fact that at maximum speed the wire-suspended models are readily somewhat disturbed and consequently give slightly lower maximum lift. This effect is probably also the cause for various identical deviations in the DVL measurements (airfoil sections 2418 and 2421).

2) With exception of sections 2418 and 2421, the extrapolation of the DVL measurements joins the results of the N.A.C.A. VDT satisfactorily so far as the VDT results for maximum pressure (effective $R \cong 8 \times 10^6$) are used. There

> the greater reliability of the N.A.C.A. VDT data is probably attributable to the extremely rigid mounting of the models on supports. The cause of the scatter of the N.A.C.A. VDT data at lower pressure is not deducible from the little available material.

It is therefore seen that the use of the term "effective Reynolds Number" is not contradictory, and its importance as criterion for applying $c_{a_{max}}$ measurements to nonturbulent flow assured. Aside from that, the data of the N.A.C.A. VDT and the DVL tunnel supplement each other so well that the $c_{a_{max}}$ curve of the three airfoil series throughout the entire practical flight range also seems assured.

Incidentally it should be noted at this point that on rectangular airfoils the measurements are fundamentally not of the $c_{a_{max}}$ of the two-dimensional problem but for slightly lower values. The divergence is due to the nonuniform lift distribution of the rectangular wing. The difference is so much greater as the flow on exceeding $c_{a_{max}}$ separates so much more suddenly. This source of error can be avoided by check tests of wings with elliptical plan form.

IV. RESULTS OF $c_{a_{max}}$ MEASUREMENTS WITH SPLIT FLAP

It was a question of extending the investigation with slotted flap or split flap. The lift increase is about the same, still the slotted flap being more commonly used because of its lower drag.

The use of the slotted flap means an added wing aside from the principal wing whose fitness is to be tested. Its use in systematic tests entails all the disadvantages accruing from the presence of a second wing, being subject to Reynolds Number and installation effects. The maximum lift increase is tied to a certain slot form and angle of slot, both of which in turn can be affected by the Reynolds Number.

The use of asplit flap means adding a baffle plate; there is no slot and the edges of separation are so clearly defined that a minimum Reynolds Number effect is expected on the flap itself. Besides, its action between 60 and 80 degrees flap angle discloses a flat optimum which minimizes the importance of flap setting as a source of error at large flap angles.

For these reasons the split flap was chosen despite the fact that the data obtained with it can be no more than approximately valid for the common split flap or the Fowler flap. It was a full-span flap with 20-percent chord, hinged at 80-percent profile chord and 60-degree setting with respect to the lower wing surface (fig. 6). On the basis of subsequent special studies it would have been better to use a 70-degree setting because it strikes the average value of the optimum angles for thick and thin airfoils more accurately, or else use an adjustable flap altogether and refer the angle of attack in proper form to the airfoil median line instead of to the pressure side. The arrangement as in figure 6 discloses - although only very little - a drawback of the thick airfoils, the thickness slightly reduces the aerodynamic angle of attack of the flap.

The maximum lift obtained with the flap arrangement of figure 6 is shown in figures 7 to 9 as a function of the effective Reynolds Number. Unfortunately the completion to $R_{effective} \cong 8 \times 10^6$ is lacking because of the absence of corresponding data from the N.A.C.A. VDT or similar tunnels. We made a temporary extrapolation on the assumption that the lift increase achieved by the split flap is not affected by the Reynolds Number, that is, we joined for the Reynolds Number range of $R_{effective} \cong 4$ to 8×10^6 the experimental curve without split flap to the test data with split flap through parallel shifting in direction of higher lift coefficients. The correctness of this extrapolation is confirmed in numerous individual split-flap tests (references 5, 9), which consistently prove that the increase in lift of the split flap is not, or only very little, influenced by the Reynolds Number.

With a few exceptions (airfoil 2409, 0012, 0015, and 23012) which again manifest a slight $c_{a_{max}}$ drop at maximum speed in the DVL tunnel, the extrapolation joins on to the test data very well. Nevertheless the extrapolation will be checked experimentally to the extent that can be achieved by addition of a turbulence screen in the DVL tunnel.

V. RESULTS OF PROFILE DRAG MEASUREMENTS AT LOW LIFT

The method of profile drag measurements developed by the DVL has been described in detail in reference 6. A résumé therefore suffices.

The profile drag measurements on the airfoil series 00, 24, and 230 were fundamentally made in two ways: first, by employing the usual method of measuring the forces on the balance, then by measuring the loss of momentum according to Betz. In these measurements the normal airfoils of 4-meter span and 0.8-meter chord were used. The fact that both methods gave the same result after rounding off the wing tips and subtracting the drag corresponding to the area of rounding, is proof that the profile drag of the plane problem had been reached very closely.

As regards the effect of the jet turbulence on profile drag, there was not and is not even today any clear perception. From comparing the DVL data with those of the N.A.C.A. VDT on the basis of the same Reynolds Number ($R_{effective} \cong 8.2 \times 10^6$) and the same tip shape (blunt tips) it may be assumed that the turbulence effect is influenced by the thickness (fig. 10). For thick airfoils the rise in profile drag due to turbulence is substantially greater than for thin airfoils. Elsewhere (reference 5) it had been attempted to convert the profile drag of an airfoil to effective Reynolds Number by subtracting the drag difference between R_{test} and $R_{effective}$ of the fully turbulent friction curve of the flat plate. The corresponding drag difference (Δc_w) has been subtracted from the two drag curves of figure 10. It is seen that, while for very small profile thickness the correction effects an approximate agreement, it is unsatisfactory for the practical range of thicknesses. So long as this difference remains to be cleared up, it is ~~mistake~~ desired even at present to make profile drag tests in low-turbulence tunnels, because they conform much better to free flight conditions.

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With low turbulence the effective Reynolds Number reached on normal airfoils in the DVL tunnel ($R_{effective} \cong 3.5 \times 10^6$) is very low compared with actual values obtained in high-speed flight ($R_{effective} \cong 10$ to 30×10^6).

So, while the low turbulence, the satisfactory design and surface of the models, and the reliability of the test method were insured, there still remained the element of doubt regarding the extrapolation to large Reynolds Numbers. In order to remove this uncertainty, a number of airfoils with unusually large chord (3.2 m) from the series 24 were investigated. The effective Reynolds Number of $R_{effective} \cong 15 \times 10^6$ obtained with them proved that the extrapolation of the profile drag in low-turbulence tunnels and with smooth airfoils approximately occurs on parallels to Prandtl's transition curve of the frictional drag of the flat plate (reference 6).

The profile drag of the N.A.C.A. series 00, 24, and 230 was determined for the plane problem at $R_{effective} \cong 3 \times 10^6$ by the described methods and extrapolated beyond the maximum test figure ($R_{effective} \cong 15 \times 10^6$) to the mean value of the practical range of Reynolds Numbers ($R_{effective} \cong 20 \times 10^6$) on the basis of the tests on the airfoils with 3.2-meter chord (reference 6). (See figures 2 and 8.) Figure 11 shows the result for $R_{effective} \cong 20 \times 10^6$ as $c_{wp}(c_a = 0.1)$ plotted against profile thickness. Usually $c_{wp_{min}}$ or $c_{wp}(c_a = 0)$ serves as reference point for the profile drag. In the present case $c_{wp}(c_a = 0.1)$, that is the profile drag for $c_a = 0.1$ is given, because it represents the mean value $c_a = 0$ and $c_a = 0.2$ conjugated to the $c_{wp_{min}}$ for 0 and 2 percent, while the value $c_a = 0.1$ itself approaches the lift values of modern high speed. The choice of c_a value for the comparison is not essential although it still has some perceptible effect when comparing closely related airfoil series.

With small thickness the profile drag of the symmetrical airfoil is superior to the two airfoils with 2-percent camber, according to figure 11. But, as the thickness increases the camber effect is neutralized by the effect of the increasing thickness.

The plotting of c_{wp} against c_a was omitted because there still exists a certain doubtfulness regarding the induced drag correction in elliptic tunnels so that the data for high c_a values do not appear as yet sufficiently safe.

VI. RECAPITULATION OF THE RESULTS

Figure 12 shows the maximum lift for the airfoil series with and without split flap for the average value of the practical range of Reynolds Numbers ($R_{effective} \cong 4 \times 10^6$) plotted against the airfoil thickness.

As regards $c_{a_{max}}$ the findings are:

1) Without split flap, airfoil series 230 is superior to the other series in thickness range of between 10 to 21 percent.

2) With split flap, airfoil series 24 gives the best results between 9- and 17-percent thickness, but for still greater thickness the symmetrical airfoils of the series 00 are superior.

A survey of the rating factor $c_{a_{max}}/c_{wp}(c_a \approx 0.1)$ is afforded in figure 13, where this factor has been plotted against the airfoil thickness with and without split flap. The mean value of the practical range of Reynolds Number was assumed at 4×10^6 for the $c_{a_{max}}$ values, and the $c_{wp}(c_a = 0.1)$ values referred to their mean value of the practical range $R \cong 20 \times 10^6$.

Regarding $c_{a_{max}}/c_{wp}(c_a = 0.1)$ figure 13 discloses the following:

1) Without split flap, airfoil series 230 (2-percent camber at 15-percent chord) is superior throughout the explored thickness range (9 to 21 percent). On series 24 the optimum $c_{a_{max}}/c_{wp}(c_a = 0.1)$ is reached at around 9-percent thickness.

2) With split flap, airfoil series 24 (2-percent camber at 40-percent chord) excels below 15-percent thickness, while the symmetrical airfoil series (0-percent camber) gives the best results when the thickness exceeds 15 percent. The optimum $c_{a_{max}}/c_{wp}(c_a = 0.1)$ for airfoil 24 is reached with 382 at approximately 12-percent thickness. A 50-percent thickness

increase lowers the $c_{a_{max}}/c_{wp}(c_a = 0.1)$ by only 5.5 percent, if at 18 percent-thickness the symmetrical airfoil is chosen.

This is proof that the introduction of landing aids shifts the rating of the airfoils considerably. The extent to which the data obtained with split flap can in principle be applied to other landing aids also remains to be proved in supplementary tests.

Translation by J. Vanier,
National Advisory Committee
for Aeronautics.

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Figure 1.- Airfoil of 4 m span and 0.8 m chord in the 5 x 7 m wind tunnel of the D.V.L.

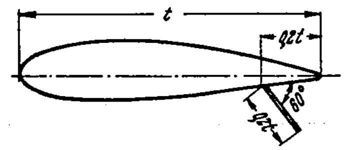
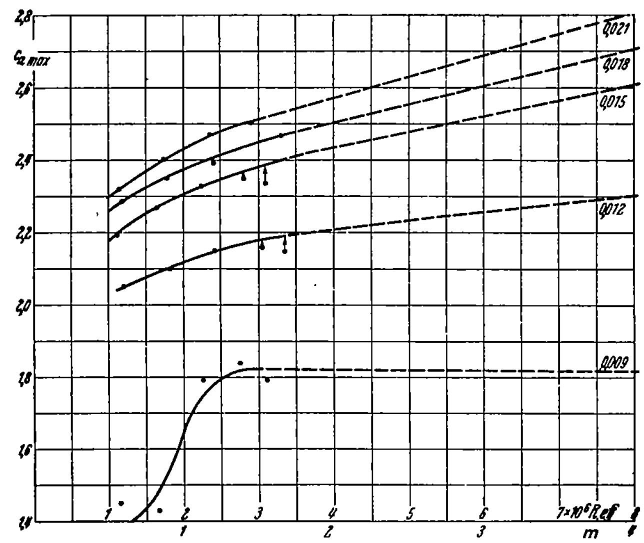
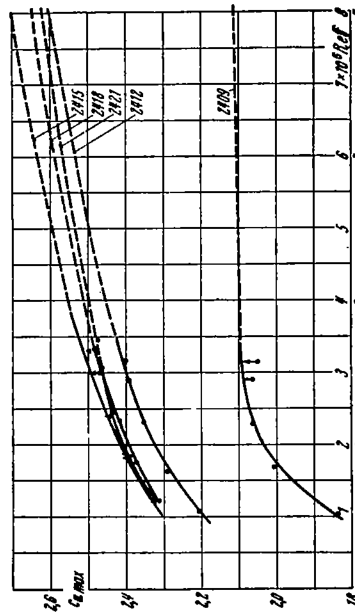


Figure 6.- Dimensions of experimental split flap.



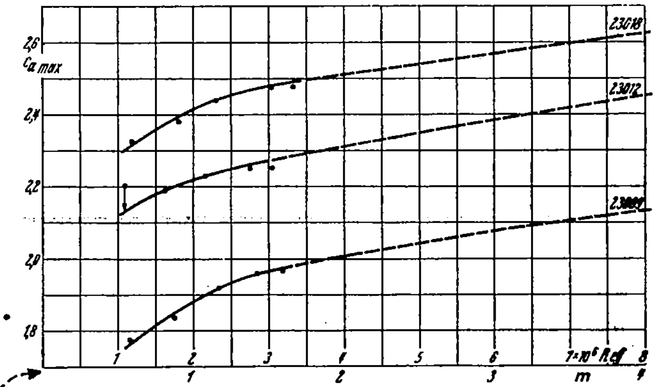
Wing chord for 100 km/h , landing speed.

Figure 8.- ca_{max} of N.A.C.A. airfoil series 0009 to 0021 with split flap.



Wing chord for 100 km/h , landing speed.

Figure 7.- ca_{max} of N.A.C.A. airfoil series 2409 to 2421 with split flap.



Wing chord for 100 km/h , landing speed.

Figure 9.- ca_{max} of N.A.C.A. airfoil series 23009 to 23018 with split flap.

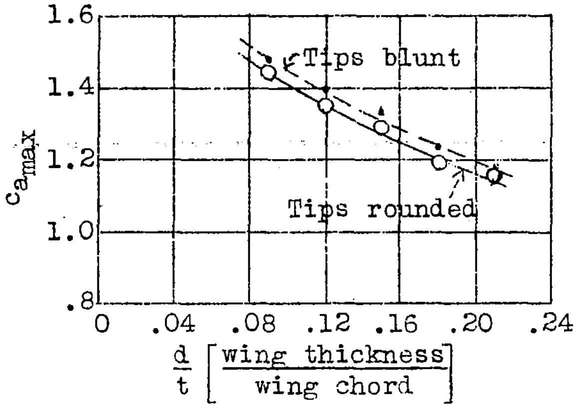
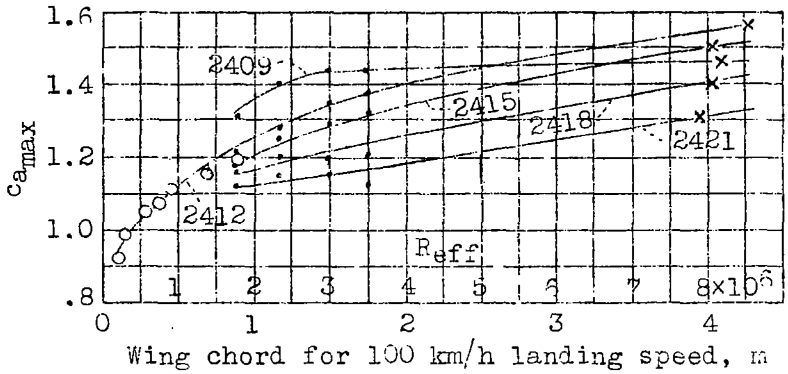
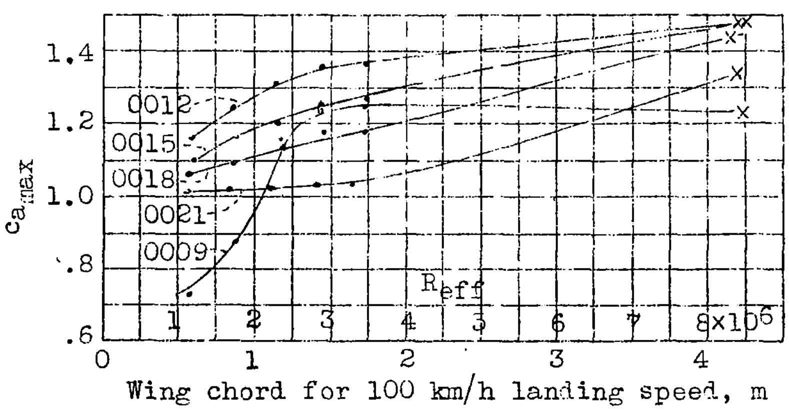


Figure 2.- Maximum lift of the N.A.C.A. airfoil series 2409 to 2421 with blunt and round wing tips. $R_{eff} = 3 \times 10^6$



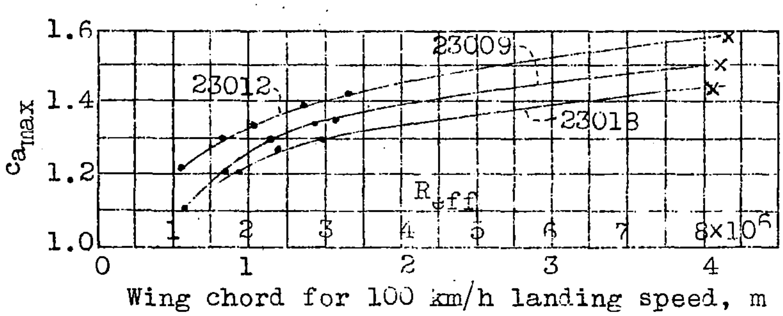
	Tunnel	Source
•	DVL 5×7m	——
o	GALCIT 3m	(7)
×	VDT 1.5m	(2)

Figure 3.- $c_{a_{max}}$ of N.A.C.A. airfoil series 2409 to 2421



	Tunnel	Source
•	DVL 5 × 7 m	——
×	VDT 1.5 m	(2)

Figure 4.- $c_{a_{max}}$ of N.A.C.A. airfoil series 0009 to 0021



	Tunnel	Source
•	DVL 5 × 7 m	——
×	VDT 1.5 m	(3)

Figure 5.- $c_{a_{max}}$ of N.A.C.A. airfoil series 23009 to 23018

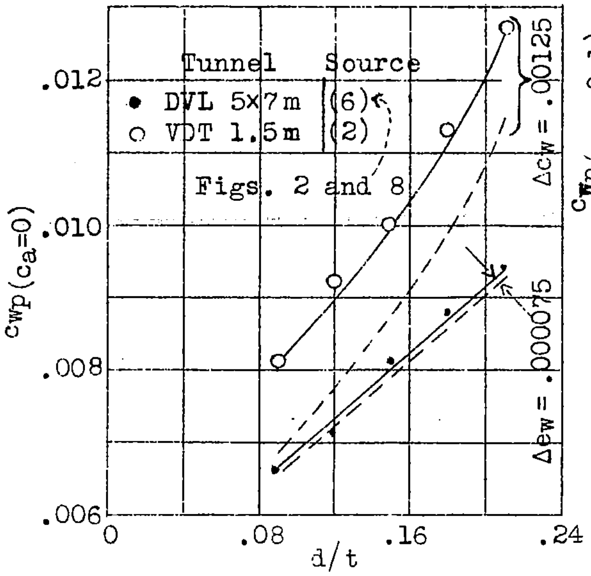


Figure 10.- Comparison of profile drag of V.D.T. and D.V.L. tunnel for N.A.C.A. airfoil series 24 on the basis of blunt wing tips and $R = 8.2 \times 10^6$

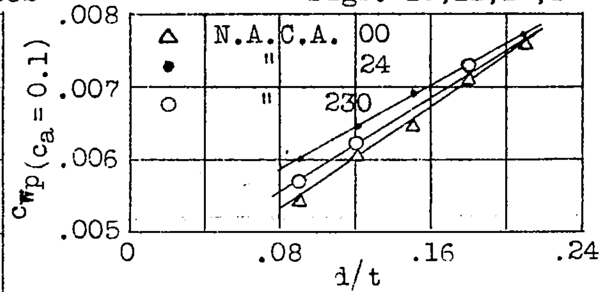


Figure 11.- Profile drag of N.A.C.A. airfoil series 00, 24 and 230 against airfoil thickness for $R = 20 \times 10^6$

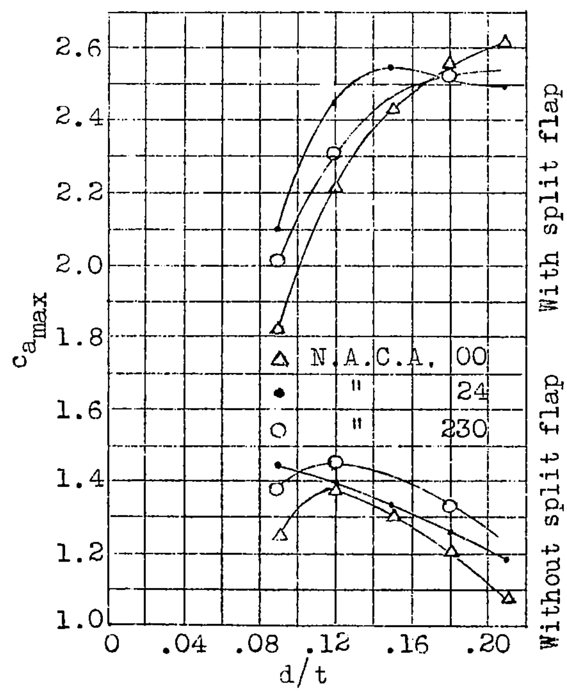


Figure 12.- $c_{a_{max}}$ of N.A.C.A. airfoil series 00, 24, and 230 with and without split flap against airfoil thickness at $R = 4 \times 10^6$

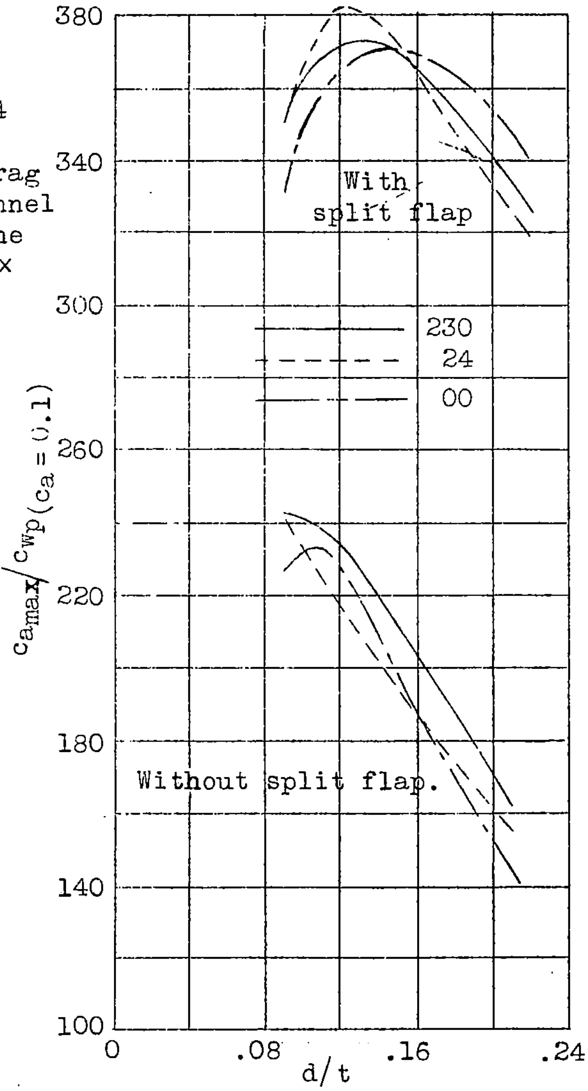


Figure 13.- Rating factor $c_{a_{max}}/c_{w_p}(c_a = 0.1)$ for N.A.C.A. airfoil series 00, 24, 230 with and without split flap against profile thickness for average Reynolds Number in free flight.